comparative Study of optical and RF Communication Systems
for a Mars Mission

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ABSTRACT

We have performed a study on telecommunication systems for a hypothetical mission to Mars. The objective of the study was to evaluate and compare the benefits that microwave (X-band and K_a-band) and Optical communications technologies afford to future missions. The telecommunication systems were required to return data after launch and in-orbit at 2.7 AU with daily data volumes of 0.1, 1, or 10 Gbits. Space-borne terminals capable of delivering each of the three data rates were proposed and characterized in terms of mass, power consumption, sire, and cost. The estimated parameters for X-band, Ka-band, and Optical frequencies arc compared and presented here. For data volumes of 0.1 and 1 Gigs-bit per day, the X-band downlink system has a mass 1.5 times that of Ka-band, and 2.5 times that of Optical system. Ka-band offered about 20% power saving at 10 Gbit/day over X-band. For all data volumes, the optical communication terminals were lower in mass than the RF terminals. For data volumes of 1 and 10 Gb/day, the space-borne optical terminal also had a lower required DC power. In all three cases, optical communications had a slightly higher development cost for the space terminal.

1. INTI{OD[JCTION

The deep space exploration program has been steadily increasing the frequencies used for planetary radio communication since the inception of NASA in 1957. L-band (900 MHz) and S-band (deep-space allocation 2,29-2.30 GHz) frequencies were used throughout the 1960's. In 1977, X-band (8.4 GHz) was put to use as the prime downlink for the twin Voyager spacecraft after experimental packages flown on the Mariner Venus-Mercury and Viking missions of the early 1970's demonstrated the viability of the frequency. In the 1990's, NASA has committed to the idea of flying smaller spacecraft with targeted science payloads. Extending the earth-space communication frequency to Ka-band (32 GHz) will allow NASA to save spacecraft mass and power while achieving the same data return volumes that X-band provides. Ka-band experimental packages on the Mars Observer, Cassini, and Mars Global Surveyor spacecraft arc helping to establish the viability of Ka-band for future deep space communicant ion uses. Use of optical frequencies is expected to afford further savings on mass and power-consumption. No deep-space (beyond the earth orbit) spacecraft has so far carried an optical communication terminal for either primary communication needs, or as an experimental payload. In recent years, there has been a growing demand for smaller-size, lower mass and lower power-consumption spacecraft subsystems and spacecrafts. This requirement has focused more at tent ion on prospects of Optical frequencies for deep-space communicant ions. System engineers for future missions arc now seriously evaluating Optical communication as a viable option for their missions. To assist mission planners, it is important to compare and evaluate the merits of telecomm systems designed for different frequencies under a common set of requirements, and for different ranges in the solar system. Ground-receiver (network) development and operation costs arc among the variables that arc important to the mission designers as well.

This paper summarizes and compares the results of telecommunication systems designed for a 2.7 AU Mars mission, The communication frequencies were X-band, KaJ3and, and Optical. Each system was required to return a daily data volume of 0.1 Gb, 1 Gb, or 10 Gb.

The requirements for the hypothetical Mars mission study were:

- Daily data volumes of: 0.1 Gb, 1 Gb or 10 Gb at 2.7 AU (nominal)
- Subsystem shall support: Telemetry, navigation, & uplink commands
- Link Availability: ≥ 95%

The study assumptions were:

- Spacecraft has on board intelligence to recover from tumbling and to rc-acquire Earth
- Three-axis stabilized spacecraft with 0.10 attitude control
- Six months at Mars, starting at minimum SEP (Sun-Earlh-Probe angle) allowed by technology
    -1° to 2° for X and Ka 10° for optical
    - Subsystem is to have 5 year life (cruise & operating)
- One week on-board data storage capacity (-9 G-bytes)
- Costs are based on 1995 dollars
- Technology freeze date of 12/1 996 at NASA Technology Readiness Level of 4*
    (* Component and/or breadboard validation in laboratory environment)

The approach taken to the study was:

- Perform conceptual designs for X, Ka and optical telecommunicat ions spacecraft subsystems
- Generate flight terminal mass, power-consurnption, and cost estimates
- Design and cost ground network to support mission
- Determine ground station and operations cost

Results of the study arc described below, and estimated parameters for each telecommunication band arc compared,

11. OPTICAL COMMUNICATIONS

11.1. Ground Receiver Network

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11.1a Subnet Concepts: Four alternate concepts were examined in this study; a seven-station optical subnet (LDOS), a three-station optical subnet in the Continental US (CONUS), and double- and single-slalion concepts. This work is based upon previously reported developments [1] and as such only ncw developments in areas of optical station design arc described in this article.

All three concepts are based upon a single station design that consists of a receiving system, transmit system, data processing and support electronics and station facilities.

Receiving System: The receiving system consists of a 10 meter non-diffraction limited segmented primary mirror with a monolithic secondary in a Cassegrain configuration as shown in Figure 11.1.1.   The telescope is mounted on an azimuth-elevation gimbal mount and is housed in a collapsible enclosure (dome). The receiving system also includes the optical bench consisting of beam reducing optics, fast steering mirror, spectral filter and tracking and communications detectors. System and component performance requirements were previously reported in Ref. 1.

The receiving aperture utilizes a segmented primary containing N segments for a total effective aperture of 10 meters. Both the segments and the aperture are hexagonal in shape.   The individual panels arc actively controlled and arc mounted on ball screw actuators. The primary support structure consists of an erectable truss manufactured from Carbon Fiber Reinforced Plastic (CFRP). The secondary mirror consists of a 1 meter monolithic primary mounted on a CFRP quadrapod. Figure 1 I. 1.2 describes the panel mounting configuration for the primary including the active control system
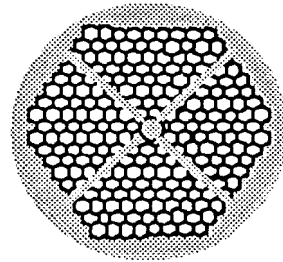
Data Processing and Support Electronics: Each optical station is designed to be fully automated and will contain 3 Sparc 20 and 2 HP workstations, a weather analysis terminal, pointing computers and controllers and a timing system.

Facilities: The facilities will consist of a 5400 square foot building housing the optics, archive and control room as well as maintenance and personnel areas. The aperture and dome will bc mounted on top of the facility. For facility related details refer to Reference 1.

Figure 11.1.1 Schematic of a 10-m receiver

Segmented Primary Mirror
Terminal Configuration

Alluminum honeycomb sandwich core

Diamond milled alluminum face

CFRP Quadrapod

CFRP Truss with Invar steel Joints

Single Secondary mirror

Primary

Secondary

Optical Bench
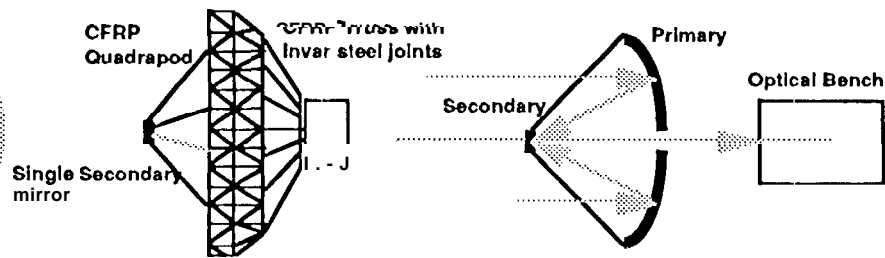
Figure 11.1.2 Panel-Mountirlg Configuration for Primary Mirror

MIRROR PANEL (SIDE VIEW)

ACTUATORS

EDGE SENSOR

CFRP TRUSS

CAPACITIVE DISPLACEMENT EDOE SENSORS

PANEL

PANEL

4 mm GAP

CAPACITORS

SENSOR FORK

70 CM EFFECTIVE DIAMETER

ACTUATION ATTACHMENT POINTS

PANEL FRONT VIEW

EDGE SENSOR LOCATIONS

PANEL BACK VIEW

DYNAMIC RANGE 25 mm
STEP RESOLUTION 0.25 um

PANEL ATTACHMENT

ROLLER NUT

MOTOR

TRUSS ATTACHMENT

ACTUATOR DIAGRAM (BALL SCREW)

II.1b Uplink Transmitter System:

A schematic of the ground transmitter station is shown in figure 11. lb-l.   11 consists of a high power laser propagated through a sub-aperture of a standard astronomical quality 1.5-m telescope. Although the telescope's principal use is to transmit uplink commands to the satellite, it is also used to acquire and track the spacecraft and to collect downlink telemetry during the transfer-orbit phase. Approaches for pointing the telescope to the spacecraft depend on the phase of the t rajectory. During the near%.arth phase telescope pointing is accomplished using a spacecraft ephemeris file that has been generated from tracking data. In the deep-space phase, guide stars arc used as pointing references [2]. 'I'his latter requirement defines the figure and hence cost of the transmitter telescope.

Because of the possible hazard posed by the uplink laser to aircraft passengers and crew, the ground transmitter station is equipped with an aircraft detection radar unit. This unit is coaligned with the beam propagation axis and is interlocked with the laser emission system. The radar system interrupts laser transmission when an aircraft is detected.

The uplink laser is based on a commercially developed diode-pumped laser technology and is a 10 Joule per pulse, 100 Hz repetit ion rate, Q-switched Nd:YAG laser operating at the 1.06 µm fundamental laser wavelength. The infrared uplink wavelengt h reduces the effects of atmospheric attenuation and scintillation, and at the same time is sufficiently removed from the 1.0S µm downlink that it provides adequate wavelength isolation between the transmit and receive beams in the spacecraft's optical train.

Figure 11.1 b-1: Schematic of the transmitter station showing the telescope control, downlink telemetry, and data flow for uplink command, Coarse pointing is accomplished using the telescope control mechani sm. A t we-axis mirror controlled through the telescope control computer is used for fine pointing adjustment of the laser beam. GCF is Ground Control Facility.
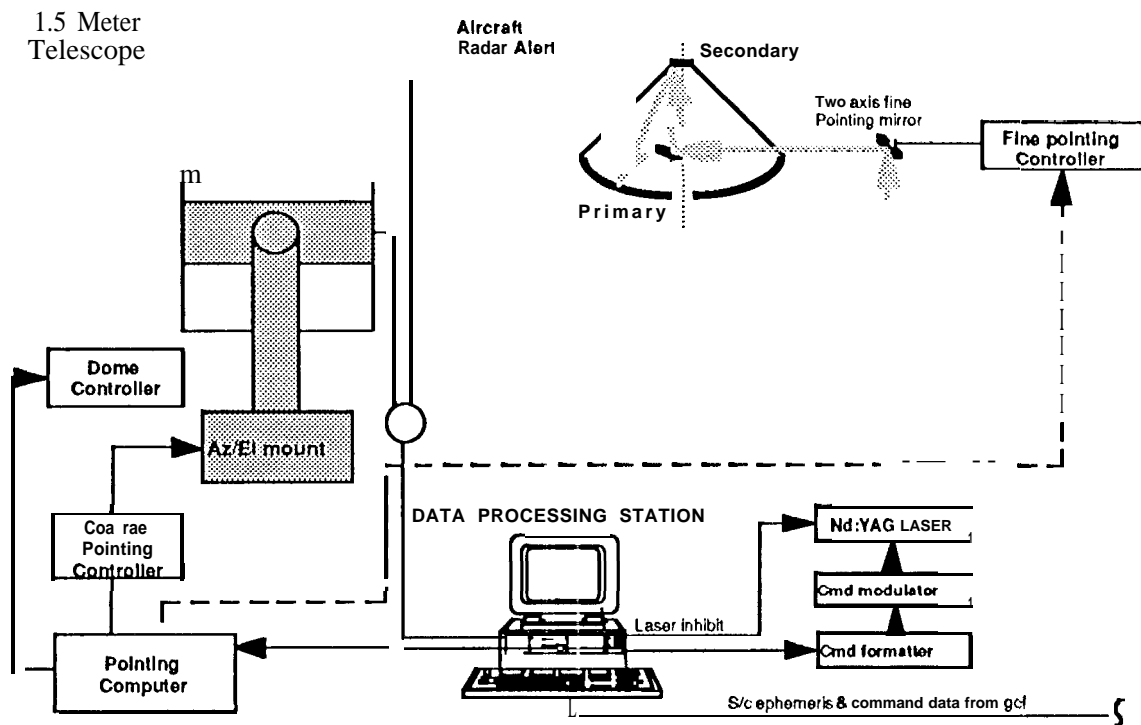
The link analysis for the uplink transmission is given in Table II-1b below. The laser beam illuminates a 20 cm sub-aperture of the telescope primary to mitigate effect of uplink scintillation and beam wander. Sub-aperture beam propagation has been successfully used in the two JPL-led laser communications demonstrations GOPEX (Galileo Optical Experiment)[3] with the Galileo spacecraft and GOLD (Ground/Orbiter Lasercomm Demonstration) [4] with the Japanese ETS-VI satellite. Transmitter stations arc located above 2 km altitude to further reduce the effects of scintillation and beam wander on the uplink. The sites arc assumed to have 4 arc seconds daytime seeing or a Fried atmospheric coherence length of 3 cm at 0.5 pm. Under these conditions the expected beam wander has a 1- sigma value ranging from 9 μrad to 21 μrad, as the zenith angle is increased from zero to 60". The link was designed with a beam divergence of approximately 35 μrad (i.e., 1.5-sigma for the beam wander at 60° zenith angle) to accommodate up to 11 dB signal fades for the 1 S cm spacecraft receiver telescope and up to 13 dB fades for the 25 cm receiver telescope.

Table II.1b. Link table for Mars day time mission

Transmitter Parameters	Value
Transmitter Average Power, W	1000
Wavelength, μm	1.06
Transmitter diameter, cm	20
Transmitter obscuration, cm	N/A
Transmission optics efficiency	0.9
Pointing bias error, μrad	5
Pointing jitter, μrad	5
Beam width, μrad	35
Operational Parameters	
Alphabet size	256
Data rate, kbps	0.6
Link distance, AU	2.7
Atmospheric transmission	0.87
BER	1E-5
Link margin, dB	
- 15 cm spacecraft receiver (0.1 and 1 Gb/day)	11
-25 cm spacecraft receiver (10 Gb/day)	13

Although, not considered in the current analysis, recent GOLD experiments [5] have demonstrated that there is a distinct reduction 'in uplink scintillation when the uplink beam is props-gated through multiple sub-apertures that are separated by greater than r_o with relative path delays greater than the laser's coherence length. A two-beam approach has been used during the GOLD experiment to demonstrate uplink bit error rates as low as 1×10^5.

II.1c Ground Network and Station Development and Operations Costs: Development and operations cots were developed first at the station level and then used as the building block for the subnets. Optical station materials and construction costs were categorized into the four main areas described above. The approximate costs for a domestic and a foreign stat ion arc $25 M and $27 M, respectively. These arc shown in Table 11-1a. The next step consisted of producing cost estimates for the development and deployment of multiple station subnets including required common support facilities such as the centralized Network Operations Control Center (NOCC). This study included 4 concepts; a seven-station.L~ a three-station CONUS network and a double/single station design. The assumed development schedule duration was 5, 4) and 3 years respectively, and would bc implemented via prime contract, Preliminary costing results for the four alternatives am: $ 272 M for the LDOS development; $120 M for the CONUS, $79 M for the double station, and $40 M for the single station alternative.

11.2 Space Terminal

The space-borne terminal consists of three main subsystems. These arc: the acquisition, tracking, and pointing subsystem; the transmitter subsystem; and the transmit/receive aperture and associated optics. Each of these subsystems is described below with major emphasis on acquisition and tracking subsystem.

11.2.1 Acquisition, Tracking and Pointing

2,1.1 a Requirements. The two basic design approaches considered were:

1. Active beacon assisted acquisition and tracking: This method uses a ground-based laser beacon to provide a pointing reference to the spacecraft terminal. Because t he beacon emanates from the receiving station, it provides a precise reference to the receiver location;

2. Earth-image-based acquisition and tracking: This method uses the sun-lit earth image as the pointing reference. The spacecraft terminal is responsible for resolving the earth image to determine the earth receiver location.
Considering operational costs and constraints, an Earlh-image-based acquisition and tracking scheme was selected for the Mars mission, beyond the link range of 0.2 AU. The implications of this choice are discussed below.

1. Focal plane array will be used for both initial acquisition and fine tracking. This design choice leads to a significant requirement on the computing power and hence a dedicated processor for the lasercom transceiver, and
2. Platform stabilization technique will be used to reduce the platform jitter on the spacecraft so that tracking loop update rate on the order of 100Hz can be employed.

Prior to establishing the communications link, the space and ground terminals need to perform mutual acquisition of each other's signal. Acquisition starts shortly after launch when the spacecraft is still in the Earth vicinity and continues throughout the mission. The acquisition process involves the following stages:

1. Determination of earth position and orientation of the spacecraft to position the Earth receiver within the field of view.

2. Identification of the position of the Earth receiver to better than 5-1 0% of the downlink signal beamwidth, and

3. Determination of the angle of the downlink beam such that the signal from the spacecraft is received by the ground station,

The spatial acquisition process is repeated after a solar conjunction outage. After acquisition has been accomplished, the two systems track each other's signal for the duration of the communication link.

Typically, the s/c beam-pointing subsystem relics on a tracking reference. The line of sight perturbation is measured using an optical detector and the measured deviation is then fed back to a beam steering device to compensate for the platform jitter.

The process of spatial acquisition and tracking includes the following factors:

1. The spacecraft attitude uncertainty and control is typically larger than the angular beamwidth of the transmit signal,
2. The random jitter aboard the spacecraft can result in a high frequency pointing jitter of the signal, The magnitude of this pointing offset is typically larger than the angular beamwidth of the transmit signal, For a spacecraft with vibration characteristics similar to that of Olympus, a tracking update rate on the order of 5-10 kHz will be required to support a 0.5-1 kHz tracking loop. A *narrower* (< 2 KHz) tracking bandwidth may be used if the s/c vibration environment is better than that of Olympus. A number of techniques including passive and active isolators can be used to reduce the frequency content of the platform jitter and hence permit a lower bandwidth tracking loop.
3. To compensate for the relative motion between the spacecraft and earth station, the transmit signal must be pointed ahead of the apparent receiver location. This "point-ahead" angle is typically larger than the transmit beamwidth.
4. Atmospheric effects can limit the use of uplink laser to provide a point ing reference. These effects include
   a. turbulence can lead to beam broadening and beam wandering
   b. turbulence induced scintillation can lead to un-predictable fades and outages, and
   c. cloud cover
5. The widely varying range (from <<0.1 AU *to* 2.7AU) leads 10a large dynamic range in uplink laser signal power and Earth image size.

The design and hence the performance of the spatial acquisition subsystem depends on a number of system parameters, most importantly, random beam steering of the downlink by the spacecraft vibration and uplink beam wander induced by the atmosphere. This induces uncertainty in illumination and can limit the tracking bandwidth.
For this study, we assumed the following:

1. Spacecraft ephemeris uncertainty during the initial acquisition period is 10-50 µrad (to be verified) if a RF tracking link is available. An alternative will be to use the Air Force Ground-based Electro-Optical Deep Space Surveillance (GEODSS) system for initial acquisition while the spacecraft is still in the Earth vicinity, GEODSS has a 4 arcsecond pixel field of view and hence a minimum initial ephemeris uncertainty of 20 µrad.
2. Spacecraft ephemeris uncertainty after solar conjunction will be less than the initial ephemeris uncertainly of 10-50 µrad.
3. The spacecraft will have sufficient attitude knowledge and control to orient the laser-conununication payload to within the field of view of the acquisition detector after a small number of search steps.
'4. The spacecraft vibration spectrum will be similar to that of Olympus spacecraft.
5. Synchronization between spacecraft and earth terminals are maintained to within a few minutes, and that the spacecraft has sufficient information to deduce the point ahead angle.

Considering these assumptions, two different spatial acquisition schemes can be evaluated, The first is an active-beacon assisted acquisition which involves the use of a ground-based laser beacon to provide the pointing reference, and the second one is an Earth-imaged based acquisition scheme in which the image of earth is resolved to provide the actual receiver location.

II.2.1a   Acquisition and Tracking rising a Ground Based Uplink Beacon

Active laser beacon has the advantage of narrow spectral width. With a narrow active spectral bandwidth filter, the laser signal can be much stronger than the background and hence can provide a higher SNR signal for acquisition. Additionally, active laser beacon provides a precise reference for the ground receiver location for initial acquisition and subsequent tracking.

The acquisition process involves the following steps:

step 1:   Spacecraft locates Earth using on-board ephemeris and star tracking to aid Earth acquisition. The s/c (or lasercom payload) then orients to position Earth within FOV of the acquisition detector and the transmit signal. This is a low risk step as technology exists today to accomplish this.

Step 2:   The ground receiver, at the pre-arranged time, will point to the general direction of spacecraft and initiate the acquisition sequence by transmitting an uplink laser beacon.   This beacon signal should be broadened to account for the possible random beam wander introduced by the at mosphere. The ephemeris error of the spacecraft is assumed to be smaller than the transmit beamwidth.

S(cp 3:   While maintaining attitude, the spacecraft lasercom terminal receives the uplink signal on the acquisition detector. The information is then used to provide a pointing reference to the transmit signal to direct a return signal to the ground station.

Step 4:   The ground receiver detects the downlink signal through an acquisition and tracking detector and uses the information to provide subsequent uplink pointing, reference to ensure proper link closure.

The risks associated with the active beacon acquisition scheme are due primarily to the attnospheric-induced beam wandering and fading. The beamwidth of the uplink signal needs to be broadened to account for the ephemeris error and the random beam wander/ beam broadening effects of the atmosphere. Additional signal margins will be required to account for the scintillation-induced signal fades which, for deep-space missions, can be as much as 10 dB as was indicated by the GOPEX data.

The required signal power and the signal to noise ratio (SNR) for tracking can be calculated based on the expected rms tracking error as follows [6]:

$$\theta_{NEA}^2 \approx \left[1.22 \frac{\pi\lambda}{4d_R} \right]^2 \frac{1}{SNR}$$

where SNR is the signal to noise ratio of the detector output over the integration period, and can be related to the detector parameter and incident signal power by

$$SNR = \frac{K_S^2}{F(K_S + K_B + \frac{I_B T_0}{e}) + \frac{I_s T_0}{eG^2} + 2\frac{k_B T_{EQ} T_0}{G^2 R_L e^2}}$$

where K_S and K_B are the detected signal and background photo-counts,
and Is are the bulk and surface leakage currents,
T_0 is the integration time,
F is the excess noise factor,
G is the APD gain,
R_L is the load resistance, and
T_{EQ} is the noise temperature of the front end amplifier.

For a tracking detector to achieve a noise equivalent angle of 1 /1 5 of a beamwidth, therefore, signal-to-noise ratio of 23 dB will be required. Note that this SNR has not taken into account the background fluctuation over the four pixels, and is significantly higher than power required for communications. Assuming an aperture size of 10 cm, an excess noise factor of 5

for the APD detector, and all background and dark currents are zero, the required signal power will be 1,000 detected photons per integration period. Considering background and detector noise, this grows to 10,000 detected photons. Or equivalently, a incident power density (for tracking) of 1.2 nW/m^2 at the spacecraft (for gated detection).

With a 10 kHz tracking bandwidth, in order to track the uplink beacon from a typical spacecraft (vibration environment similar to Olympus), a 49 kW (equivalent cw) laser with a 10 μrad uplink beam-width is required. Furthermore, the pointing error of the uplink should be less than 1 μrad (1 sigma). This is a stringent demand on the uplink, especially when daytime uplink is required during which no visible guide-stars can be used to calibrate mount distortion and atmospheric bending. Even when operating at night time, the system may still suffer from the bending mode (tip-tilt modes) of the atmosphere unless adequate calibration (such as the use of a natural guide-star or the s/c downlink signal) can be used,

With a 10 KHz tracking bandwidth, it is easy to argue that a beam-based tracking system will have difficulty satisfying the pointing requirement. The required power can be reduced by lowering the required tracking bandwidth. If the tracking bandwidth can be reduced to 100 Hz with the use of a vibration isolation stage, the required signal power (0.5 kW) is within the achievable technology. An SNR of 23 dB was assumed for tracking.

II.2.1b   Acquisition and Tracking using Sun-Lit Earth

An alternative option for spatial acquisition and tracking is to track the sun-illuminated Earth as the beacon source. The spacecraft system in this case simply determines the pointing orientation by inspecting the image of Earth and uses that information to direct the return signal without the need of an uplink beacon laser, The Earth image provides ample signal power for accurate tracking.

The Sun-lit Earth acquisition and tracking process is divided into a number of steps:

Step 1.   The spacecraft, using onboard ephemeris and star trackers, orients the lasercom payload such that the image of Earth falls onto the acquisition detector.

Step 2.   A wide field of view tracking detector (possibly a quad APD) is used to lock onto the Earth image, thereby stabilizing the image on the acquisition detector focal plane. The tracking detector maintains the receive line d sight along the direction to the Earth image centroid.

Step 3.   The image detected by the acquisition detector, which is typically a partially illuminated Earth, is processed to deduce the ground station location using information on local time at the receiver, the sun-%lh-probe geometry, and the spacecraft ephemeris. Because of the albedo variation, processing using invariants such as the shapes of the Earth limb is preferred over simple correlation.

Step 4.   Once the location of the receiving terminal is deduced from the image data. the offset between the ground receiver and image centroid is then calculated.

Step 5.   The offset angle calculated above and the point ahead angle arc used to drive the point ahead mirror and to maintain the return signal on the Earth terminal.

Step 6.   Periodically, step 3-5 is repeated to provide an update of the receiver location, which may be changing due to earth rotation.

The steps described above can be used to accurately deduce the receiver location and to stabilize the pointing based on the focal plane image. However, there are several constraints associated with this extended source acquisition and tracking algorithm:

1.   Near Earth Acquisition: During the initial acquisition, the spacecraft can be relatively close to Earth and therefore the angle subtended by the Earth image can be larger than the field of view of the tracking and acquisition detectors. This may cause problem since there will be insufficient information for tracking and acquisition processing (without data on the earth limb, for example).   For very close-in period, therefore, a beacon-based acquisition and tracking will have to be used.

2.   Image Scaling: The size of the earth image will vary when the distance to earth vanes. For spacecraft at 0.2 to 2.7 AU, the size of image can vary by as much as 13.5 times.   This leads to a very large tracking array requirement. Furthermore, the required processing power increases as the size of the earth image increases. One way to reduce the array size and the processing power growth is to use a zoom element to maintain earth image at constant size. Another approach is use of a quadrant detector in the focal plane for tracking.

3.   Albedo fluctuation: Variation in Earth albedo can cause false edge detection during the limb determination process and lead to error in determining the receiver location.   Variation in the receive position estimates have been estimated

using a Monte-Carlo simulation, and the results generally stales that albcdo-induced variation can be as large as 1/1 O of the beamwidth (1 sigma). Further refinement of the algorithm is needed to reduce the error in this step.

4. Processing speed: The image processing will impose serious constraints on the speed of the processor board, Signal processing schemes previously proposed for extend source tracking require extensive signal processing that is unlikely to be available with the spacecraft CDH. Consequently, a dedicated processor will have to be employed.

5. Platform Isolation: Even with a dedicated processor an isolation stage will be required to reduce the required update rate.

6. Near Sun Acquisition and Tracking:The extended source acquisition scheme relics on the Earth image which is generally in the same spectral band as the solar background. Consequently, conventional background rejection techniques (narrow-band filtering) cannot be applied. The near sun acquisition constraints can lead to communication outages during superior conjunctions.

Despite the large number of constraints, the Earth image based acquisition and tracking is superior compared to the active beacon based acq & track scheme for the following reasons:

1. It is feasible for missions at longer ranges whereas an active beacon-basrxi system is only applicable to shorter range missions,

2. It is more robust and less sensitive to scintillation outages.

To employ the earth image tracking concept, however, will result in the following impact on the system design

a. The optical system will likely to require a variable focal length to accommodate near Earth tracking Alternatively, the optical system will require switchable narrow-band filter and sufficiently intelligence to switch between beacon 1 racking and earth tracking.

b. Tracking directly off CCD images cannot provide sufficient tracking bandwidth unless (a) an isolation system is imposed, or (b) an auxiliary tracker (quad APD, for example) is used to track the image centroid.

c. Processing of the CCD image to retrieve pointing information will require extensive CPU power that can only be met with a dedicated processor for the lasercom system.

d. Spacecraft will need sufficient intelligence to predict the relative geometry of the sun Earth probe system

Risk Areas:

a. Image albcdo variation can distort the pointing estimate derived from the Earth image. The impact of this will need to bc assessed further.

b. Mechanism of switch-over from Near Earth (0.2 AU) tracking.

II.2.1c   Other Acquisition Methods:

Other acquisition methods such as acquiring the sun or a reference star arc not very feasible. For sun acquisition, the sun subtends a large enough solid angle (0.3 degrees at 1.5AU), that deriving accurate pointing information down to microradians using sun acquisition is very difficult. Acquisition of other stellar objects such as planets, or asteroids arc a possibility. Here at least two objects are used to determine the Earth terminal location. A wide field of view (>0.5 degree) detector with a large dynamic range is needed to image the two sources and to derive the Earth location.

11.2.1 d Conclusion. Extended-source tracking using the sun-lit Earth is a viable option for noisy s/c platforms. This option was selected for the present study. However, for the Mars mission (2.7 AU), tracking an uplink laser beacon is a viable option also. It would be considered for the follow-on studies since it eliminates the need for a dedicated flight processor.

The usc of a earth image-assisted acquisition imposes certain constraints on the spaceborne lasercom terminal. These constraints arc previously described and will be summarized here:

a, The need to incorporate a zoom element in the optical design to maintain constant image sire, or usc a quadrant detector in the focal plane

b. An effective isolation system to reduce the tracking bandwidth to approximately 100Hz.

c. A focal plane array with sufficient number of pixels and

d.   A dedicated processor for processing the focal plane image to provide image motion stabilization and to resolve the earth receiver location.

11.2.2. Space-borne Terminal

11.2.2a Design Considerations.   Emphasis was on reduced mass and power consumption for the space-borne terminal. Briefly, the design uses internal steering elements for beam pointing stabilization, an onboard star-tracker for initial attitude determination, and a combination of platform isolation and active optical feedback of sun-lit Earth images for jitter compensation. Also, The. terminal uses precision two-way ranging to derive necessary navigation variables, dual-redundant active optical elements for improved system reliability, and on-board spacecraft intelligence for attitude recovery and safe-mode operation,

11.2.2b System Design.   A block diagram of the optical transceiver and its interface with the s/c, along with a schematic diagram of the hardware are illustrated in Figures 11.a and 11.b, respectively.   Redundant units on the transceiver are shown as shaded boxes. The spaceborne terminal consist of a single receive-transmit telescope where receive/transmit isolation is obtained by proper filtering of the diverse wavelengths involved. The optical train consists of two main arms: a transmit arm which originates at the laser transmitter and extends to the telescope, and a receive arnr including data-reception and acquisition and tracking which originates at the telescope and terminates at the focal plane detectors. The two arms overlap each other during part of their paths. This ensures that the transmit beam is pointed at the ground-based beacon (which may be the sun-illuminated earth). Redundant laser transmitters are diode-purnped Nd: YLF lasers (at 10S4 nm) that provide the peak powers that are necessary for communications from the range of 2.7 AU. To reduce system mass, where possible, the optical train of the transceiver was designed based on bulk-optics.   The focal plane of the receive/transmit telescope was assumed to consist of two APD data detectors and two active-pixel-sensor (APS) detectors for acquisition and tracking, all tightly packaged. In this design a zoom lens was incorporated into the optical train to change the image size on the focal plane. As discussed earlier, another approach may use a quadrant tracking detector in the focal plane. The second approach may actually reduce mass and power-consumpt ion and increase reliability, and will have to be explored in future designs. As shown on the block diagram (Figure 2a), the transceiver package includes a Laser-conununication Control Unit (LCU). LCU is a processor (e.g. Motorola's MOPS6000) in contact, through the spacecraft bus, with spacecraft's Command and Data Handling (C&DH) unit.   Its functions are to process acquisition and tracking data (e.g., for extended-source tracking), transmit data, and ranging data for navigation.

11.2.2c Link Control Table.   Link control tables for the three data volumes of 0.1, 1, and 10 Gbits per day are summarized in Table 2.2-1 These were obtained using JPL's OPTI program. The analysis assumed a maximum range of 2.7 AU, required bit error rate of 1x10-5 (with coding), at mospheric transmission factor of 0.77 al 1054 nrn, a 10 meter photon-bucket receiver, day-time as well as night-time ground reception, and a link margin of 3 dB. Analysis indicated that a 1.35 W (average power) laser transmitter in conjunction with a 25-cm transmit aperture will provide a data rate of 200 kbps and data volume of 10 Gb/day. The required aperture size and laser output power decrease to 10 cm and 0.45 W , respectively, for daily communication volume of 0.1 Gb.

11.2.2d Mass, Power, Size, and Cost Estimates.   Mass, power, sire, and cost estimates for the spxc-borne terminal are summarized in Table 2.2-2. To calculate mass and power consumption, the system was partitioned into subsystems and then into individual components. A mass and power consumption value was allocated to each component and/or subsystem. Cost estimates for the first flight unit were calculated in the same manner.   A major (>50%) reduction in the cost of subsequent units is anticipated, Costing assumptions were: all costs in constant 1995 dollars and technology freeze date of 12/1996.   Cost includes: system engineering; flight hardware and software development; ground support equipment (for testing); system integration and test support; mission operations planning support; reliability engineering and quality assurance; and management.   Deliverables are: development and breadbording of critical components, namely, acquisition and tracking focal plane, and lasers; delivery of one flight unit plus documentation; and two sets of ground support equipment (1 set delivered to launch site).   No technology development cost included other than stated above. Development cycle is 3 years from start to launch

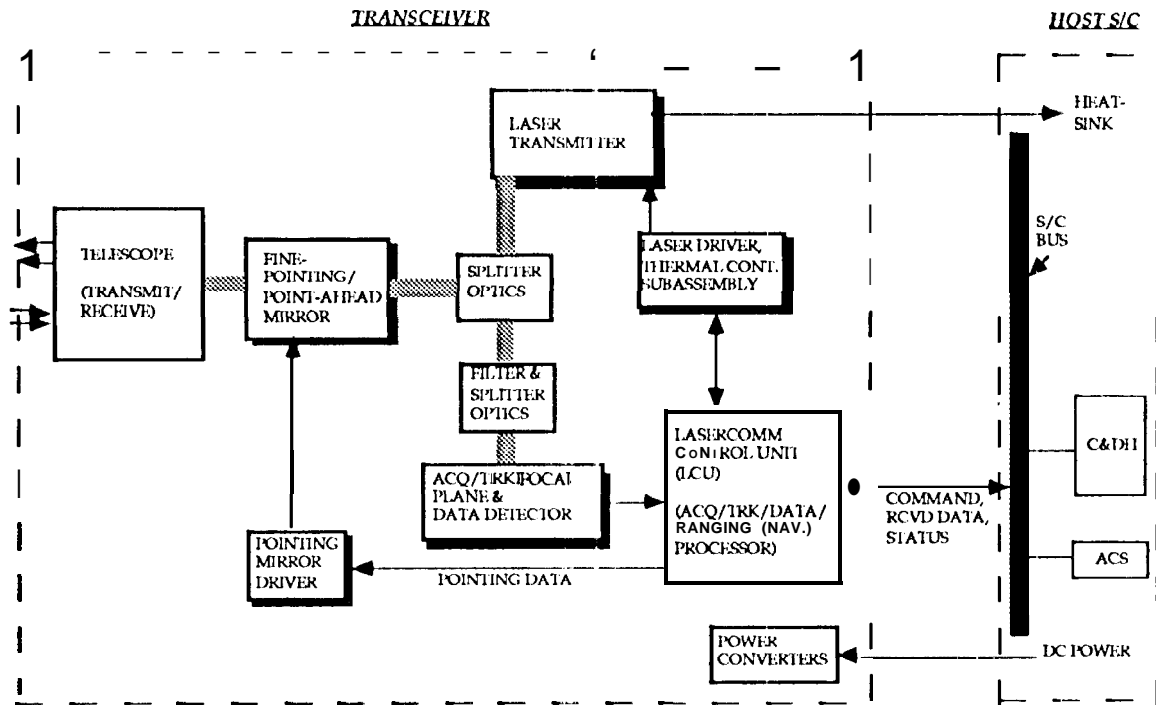**Figure ha.** Block Diagram and Spacecraft Interface

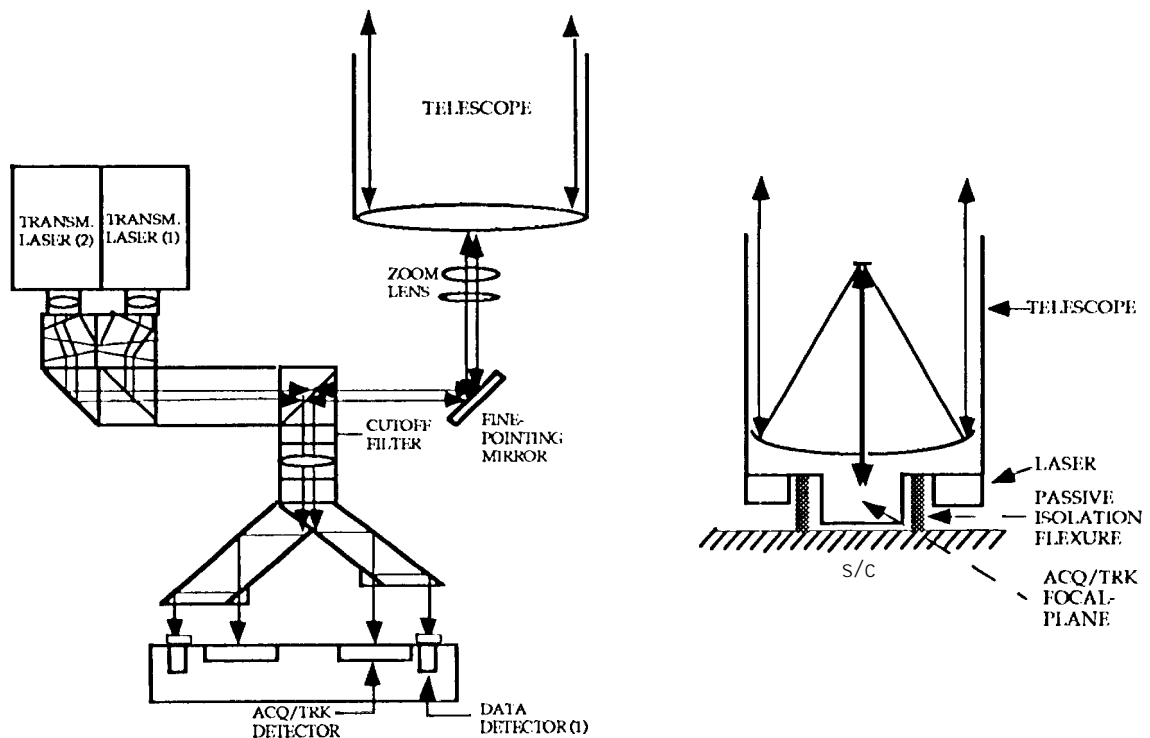**Figure II-b.** Schematic of the transceiver

Table II.2-1 Link Control Table Summary

INPUT/OUTPUT PARAMETERS	Data Volume*		
	0.1 Gb/Day	1 Gb/Day	10 Gb/Day
Data Rate (Kbps)	15	50	200
Diameter of Transmit Telescope (cm)	10	15	25
Average Output Power (W)	0.45	0.69	1.35
Ground Network Contact (hr/day)	1.8S	5.55	13.0
Beamwidth (μrad)	)7.5	12	7.0
Required Pointing Accuracy (μrad)	1.7	1.2	0.7

* Assuming: Transmit laser wavelength of 1054 run; lk distance of 2.7 J; ground receiver diameter of 10 m; required BER (with 3/4 coding rate) of 1E-5; atmospheric transmission factor of 0.77 (at 20° elevation angle); and link margin of 3 dB.

Table 11.2-2 Mass, Power, Size, and Cost Summary

	Data Volume		
	0.1 Gb/Day	1 Gb/Day	10 Gb/Day
Mass (kg)	6.4 ± 0.9	7.4 ± 1.2	9.0 ± 1.2
Laser Subsystem (Redundant)	2.1	2.5	3.1
Transmit/Receive Aperture	1.3	11.9	2.8
Acq/Trk/Ptg/Com Subsystem (Redundant)	0.7	0.7	0.7
Thern~l/Mechanical	0.8	0.8	0.9
Other (Processor, . ..)	1.5	1.5	1.5
Power (W)	19.9 ± 7	22.8 ± 4	30 ± 0.6
Laser Subsystem	4.7	7.2	13.7
Transmit/Receive Aperture	0.5	0.5	0.5
Acq/Trk/Ptg/Com Subsystem	3.6	3.6	3.6
Thernwl/Mechanical	0.5	0.5	0.5
Other (processor, . ..)	10.6	11	12
First Flight Unit (NRE + RE) Cost ($ M)	14.2	14.7	15.5
Size (cm³)	15x15x30	20x20x40	30X30X45

II.2.2e improvement Areas. Reductions of system mass and power consumption for the space-borne system arc possible in the following areas: acquisition and tracking subsystem; laser transmitter subsystem; and mission design. Development of sunlit-Earth acquisition and tracking algorithm that is less processing intensive than what was assumed here could reduce power consumption by as much as 6 W. Development of the technology of laser transmitters with emphasis on higher power efficiency (relative to the current typical value of 10OA) will lower systems mass and power consumption. Finally, planning the mission to maximize use of night-time ground reception will reduce system's mass and power requirements.

111. X-BAND and Ka-BAND COMMUNICATION

111.1 RF Design Methodology

A spacecraft radio communication system supports command, telemetry and radiometric tracking (navigation) functions. Since the downlink data rate requirement for the return of science and navigation digital imaging is much greater

than the uplink requirement for commands to the flight computer, the downlink data rate drives the system design, The design approach followed in this study was 10 minimize total impact of the telecom subsystem by minimizing the radio system mass and power consumption. The rigid parabolic reflector antenna and transponders comprise most of the system mass, while the RF transmitter consumes the majority of power. A database containing several sizes of antennas and power amplifiers was established. These candidate subsystems were taken from a combination of heritage designs and concept designs for other missions, including Cassini, Mars Pathfinder, Pluto Express, and the Space Infrared Telescope Facility. Subsystems were selected from the database to create a design which would satisfy the link requirements. X-band (8,4 GHz) was used for tbc uplink in all designs. Two different downlink frequencies, X-band and Ka-band (32 GHz), were examined as shown in Table 111-1.

To meet the daily volume requirements of 0.1, 1, or 10 Gb pcr day, the RF link design assumed onc track of a NASA Deep Space Network 34 meter antenna was available. A worst case Earth-Mars range of 2.7 astronomical units (AU) was used, as would bc seen during the Mars superior conjunction of August 2002. The downlink channel uses a (15,1/6) inner convolutional code concatenated with a Reed-Solon~on (255,223) outer code requiring a 0.3 dB-Hz bit signal to noise ratio at the input of the dccodcr.

Table 111-1. RF Downlink Budgets for 1 Gb/day Data Volume

	X-band (8.4 GHz)	Ka-band (32 GHz)
RF power, dBW (W)	7.0 (5)	5.4 (3.5)
Circuit loss, dB	-0.s	-0.3
Spacecraft antenna gain, dBi	42.7 (2.0 m)	50.8 (1.4 m)
Spacecraft pointing loss, dB	-0.1	-0.5
Space loss (2.7 AU), dB	-283.1	-294.7
Atmospheric attenuation, dB	-0.1	-0.4
Ground pointing loss, dB	-0.1	-0.3
G/T, dB/K	54.3	61.4
Boltzman's constant, JIB-J/K	-228.6	-228.6
Total power to noise, dB-Hz	48.7	47.8
Data Rate, kbps (dB-Hz)	47 (46.7)	58 (47.6)
System loss, dB	-0.4	-0.4
Bit signal to noise Eb/No, dB-Hz	1.6	2.0
Required Eb/No, dB-Hz	0.3	0.3
Link margin, dB	1.3	1.7

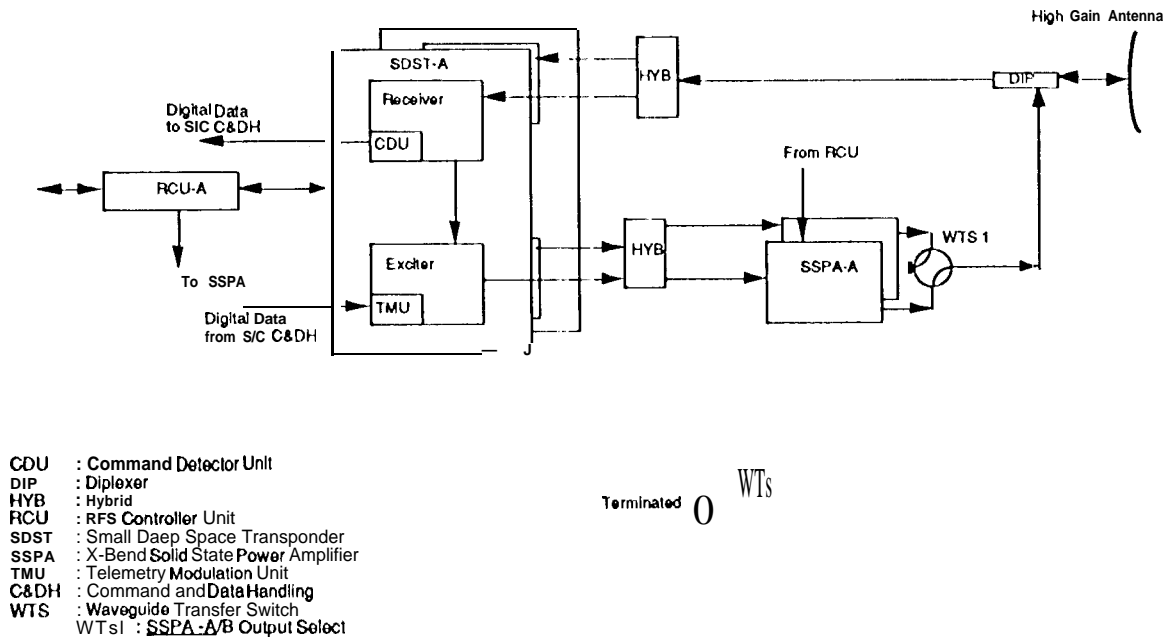## 111.2 Radio Frequency (RF) Telecommunications Hardware Systems

The RF telecommunications system is usually comprised of three major subsystems: Antenna, High Power Amplifier (I I PA), and a Transponder. Each of these subsystems have several major components. For example, the antenna subsystem has at least a High Gain Antenna (HGA) and a Low Gain Antenna (LGA) for emergency or uplink purposes. The transponder subsystem consists of a receiver and an exciter, and the High power amplifier can bc a Traveling-Wave Tube Power Amplifier (TWTA), or a Solid State Power Amplifier (SSPA). For the purpose of this study, t\vo RF telecommunications systems options were evaluated, the first was an X/X system and the second in an X/Ka system. For both systems, the LGA is omitted. Several assumptions arc made for both X/X and X/Ka system: onc High Gain Antenna (HGA) and connecting waveguide and redundant transmitters and transponders. A database is established which contains data for a variety of components some of which arc inherited from other projects such as, Cassini, Pluto Fast Flyby (PFF), Mars Pathfinder (MPF), and Space Infrared Telescope Facility (SIRTF), while the remainder arc from manufacturers data for space components. In Section 111.2.1 we begin by presenting, the X/X Systems followed by the X/Ka systems in Section 111.2.2, and finally in Section 11.2.3, the comparison between X/X and X/Ka band systems and a summary of RF system results will bc presented.

]11.2.1 X/X RF Telecommunications Systems
A schematic of the X/X RF System is shown in Figure 111.2.1. For the three data volumes, 0.1, 1.0, and 10.0 Gbits/day, the major system components arc an HGA, Diplexer, Redundant Small Deep Space Transponders (SDST), redundant X-band I] PA, Hybrids, and a Waveguide Transfer Switch (WTS). The physical characteristics for each of the

above components arc shown in table 111.2.1. All of the hardware components arc the same for three data volumes except for the }] PA. For the 0.1 Gbits/day, two 0.5 Watt X-SSPAS are used, each weigh 0.07 kg and consume 1.4 Watts. For 1.0 Gbits/day, two 5 Watt X-SSPAS arc used, each weigh 0.17 kg and consume 15 Watts. For the 10.O Gbits/day, two 22 Watt X-band TWTAs (Hughes Skynct) arc used, each weigh 3.44 kg and consume 48 Watts. All the three designs are based on a HGA of 2.0 meter in diameter, which has a mass of 7.9 kilograms including a connecting waveguide. Also the SDST includes the Command Data Unit (CDU) and the Telemetry Modulation Unit (TMU). The Diplexer design is inherited from the MPF project, the WTS design is inherited from the Cassini project, and the Radio Frequency System Control Unit (RCU) design is inherited from the PFF study. The total mass and total power consumption for the three data volumes arc shown in table 111.2.1.

Figure 111.2.1 X/X System Design, No LGA

CDU   : Command Detector Unit
DIP   : Diplexer
HYB   : Hybrid
RCU   : RFS Controller Unit
SDST   : Small Daep Space Transponder
SSPA   : X-Bend Solid State Power Amplifier
TMU   : Telemetry Modulation Unit
C&DH   : Command and Data Handling
WTS   : Waveguide Transfer Switch
       WTsl : SSPA -A/B Output Select

Terminated O WT_S

Table 111.2.1 X/X System Hardware Description List,
Data Volume 0.1, 1.0, and 10.0 Gbits/day

No. of Units	Data Volume Mbits/day	Item	Unit Mass (kg)	Total Mass (kg)	Volume (cm	DC Power (Wat t)
1	0.1, 1.0, 10.0	HGA	7.9	7.9	2-m diameter	
1	0.1, 1.0, 10.0	Diplexer	0.55	0.55	31x18x5.4	
2	0.1	0.5 W X-SSPA*	0.07	0.14	7.6x4 .5x2.5	1.4
2	1.0	5 w x-S SPA	0.17	0.34	9.8x8 .8x2.5	15
2	10.0	22 W TWTA (Hughes Skynct)	3.44	6.88	53x13 .9x24.3	48
2	0.1, 1.0, 10.0	SDST	1.8	3.6	10x 10x8	11
1	0.1, 1.0, 10.0	Waveguide Switch	0.4	0.4	10x8x5.4	
1	0.1, 1.0> 10.0	RFS Control Unit	1.5	1.5	18x10x2.5	3
2	0.1, 1.0, 10.0	Hybrid	0.1	0.2	5x5.6x2.5	
1 Lot	0.1, 1.0, 10.0	Waveguides/Mist.	2,2	2.2	2.5x2 .5x1O	
Total	0.1			16.48		]5.4
	1.0			16.68		29
	10.0			23.23		62

* The series of X-band SSPA'S (0. SW, SW) is based on the Jet Propulsion Laboratory in-house design experience of the MPF 13W SSPA.

111.2.2 X/Ka RF Telecommunications Systems

A schematic of the X/Ka RF System is shown in Figure 111.2.2. For the three data volumes, 0.1, 1.0, and 10.0 Gbits/day, the major system components are an HGA, Filter, Redundant Small Deep Space Transponders (SDST), redundant X-band HPA, Hybrids, and a Waveguide Transfer Switch (WI'S). The physical characteristics for each of the above components are shown in table 111,2.2. The first two designs are based on the use of a 1.4 meter diameter HGA, that weighs 2.5 kilograms and includes a connecting waveguide. These two designs use Redundant Ka-band SSPA's of different power levels, 0.5 W and 3.5 W for the data volumes of 0.1 and 1.0 Gbits/day respectively. The third design (10 Gbits/day) is based on a 2.0 meter diameter HGA, that weighs 7,9 kilograms and includes the connecting waveguide. Redundant 15 W T WTA's are used for HPA's, each weighs 3.44 kg and consumes 32 Watts. The TWTA design is inherited from the Cassini project. Also the mass of the SDST includes the Command Data Unit (CDU) and the Telemetry Modulation Unit (TMU). The 1.4 meter HGA design is identical to the SIRTF HGA, while the 2.0 meter HGA and the Radio Frequency System Control Unit (RCU) designs are inherited from the PFF study. The filter is the Mars Global Surveyor (MGS) project design. The WTS is a spare from the Cassini project. The total mass and total power consumption for the three system designs corresponding to the three data volumes are shown in table 111,2.2.

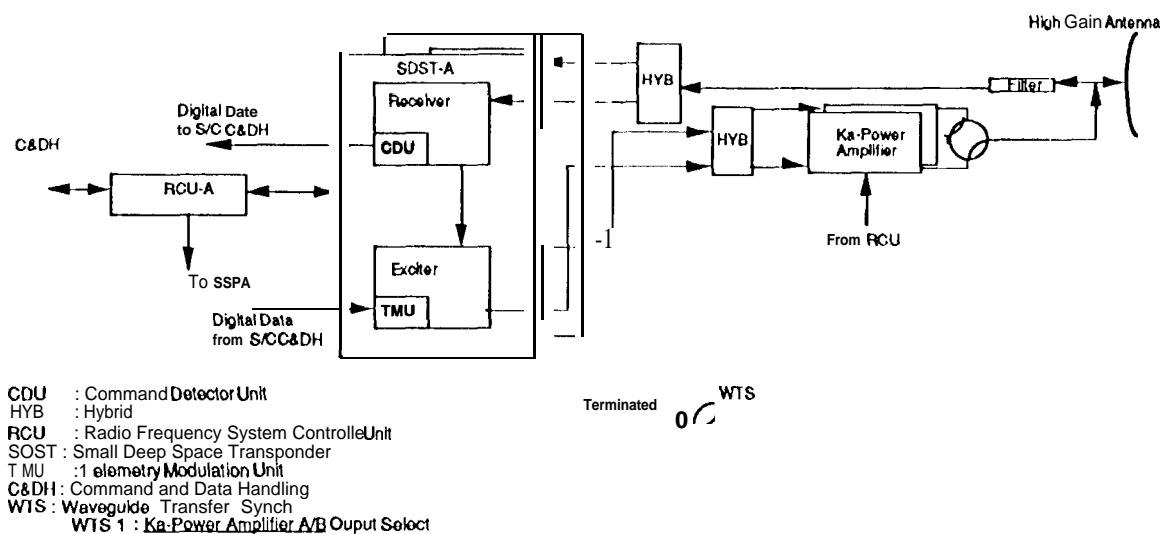**Figure 111.2.2.** X/Ka System Design, No LGA, with X/Ka Feed

CDU : Command Detector Unit
HYB : Hybrid
RCU : Radio Frequency System Controlle Unit
SOST : Small Deep Space Transponder
T MU :1 elemetry Modulation Unit
C&DH : Command and Data Handling
WTS : Waveguide Transfer Synch
    WTS 1 : Ka-Power Amplifier A/B Ouput Seloct

Table 111.2.2. X/Ka System Hardware Description List (Data Volume 0.1, 1.0, and 10.0 Gbits/day)

No. or Units	Data Volume Gbits/day	Item	Unit Mass (kg)	Total Mass (kg)	Volume (cm	DC Power (Watt)
1	0.1, 1.0	HGA	2.s	2.5	1.4 m diameter	
1	10.0	HGA	7.9	7.9	2.0 m diameter	
1	0.1, 1.0, 10.0	Diplexer	0.55	0.55	31x18x5.4	
2	0.1	0.5 W Ka-SSPA	0.07	0.14	7.6x4 .5x2.5	1.8
2	1.0	3.S W Ka-SSPA	0.17	0.34	10.4x9.4x2.5	18
2	10.0	15 W Ka-TWTA	3.05	G.1	34x22x22	32
2	0.1, 1.0, 10.0	SDST	1.8	3.6	10xIOX8	11
1	0.1, 1.0, 10.0	Waveguide Switch	0,4	0.4	10x8x5.4	
1	0.1, 1.0, 10.0	RFS Control Unit	1.5	1.5	I8x1OX2.5	3
2	0.1, 1.0, 10.0	Hybrid	0.1	0.2	5x5.6x2.5	
1 Lot	0.1, 1.0, 10.0	Waveguides/Misc.	2.2	2.2	2.5x2.5x10	
Total	0.1			11.09		15.8
	1.0			11.29		32
	10.0			22.45		46

111.2.3 Comparison of X/X and X/Ka Band Systems

A comparison of the total mass, cost, and power consumption for the X/X and X/Ka systems is shown in tables 111.2.3. The X/Ka system offers significant mass savings of up to 5 kg over the X/X system at all data volumes, and significant power savings of 16 W at the 10 Gbit/day data volume.

Table 111.2.3. Total Cost of X/X and X/Ka Systems

	Frequency Band	0.1 Gbits/day	1.0 Gbits/day	10.0 Gbits/day
Cost, $K	X/X	10164	10514	11264
Mass, Kg	X/X	16.5	16.7	23.2
Power, Watt	X/X	15.4	29	62
Cost, $K	X/Ka	10864	12289	12114
Mass, Kg	X/Ka	11.1	11.3	22.5
Power, Watt	X/Ka	15.8	32	46

### 111.3 Ground Antenna Description

The ground antenna systems used for the mission set comparison arc the 34 meter diameter X-band high-c fficiency (HEF) antenna, and the new 34 meter diameter beam waveguide (BWG) antenna currently under construction for the Deep Space Network. The BWG antenna, shown in Figure III-3, is capable of simultaneous X/Ka uplink/downlink operation through the usc of a dichroic plate Diplexer. The microwave receiver packages at both antennas consist of cooled high electron mobility transistors (HEMTs). The expected G/T performance values for these antennas are given in Table III-3 below.

Table 111-3. Expected HEMT Temperature and G/T of 34m DSN Antennas

	X-hand 34m HEF	Ka-band 34m BWG
1 I EMT Temperat u re (K)	6	13
G/T (dB)	54.3	61.4

Note: assumes 30 degree elevation angle and 90% weather

Figure 111.3. Schematic of the 34-m Beam-Waveguide Antenna
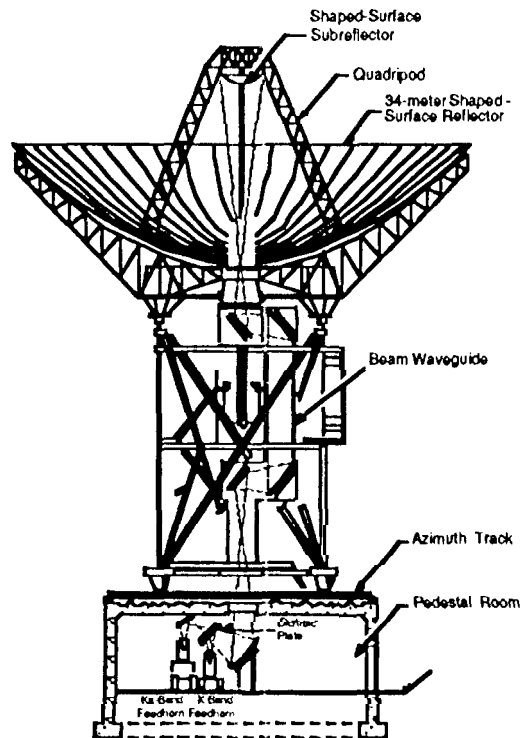
IV. SUMMARY OF FLIGHT TERMINAL RESULTS

Comparison of the results for the three telecomm systems and for the three data volumes is presented in Table IV-1 and conclusions arc given below.

Table IV-1. Comparison of the Three Communication Systems at Three Different Data Volumes

Data Volume (Gb/day)	Communication Band	Mass (kg)	D.C. Power (W)	Cost ($ M) (first unit)
0.1	X-Band	16.5	15.4	10.1
	Ka-Band	11.1	15.8	10.9
	Optical	6,4	19.9	14.3
1.0	X-Band	16.7	29	10.5
	Ka-Band	11.3	32	12.3
	Optical	7.4	22.8	14.7
10	X-Band	23.2	62	11.3
	Ka-Band	22.55	46	12.1
	Optical	9	30	15.4

1) 0.1 to 1 Gb/day. For daily volumes of 0.1 Gb and 1 Gb, conclusions were:
- Optical communication flight unit consistently requires lower mass. X-X telecom is 1.5 times mass of Ka, and 2.5 times mass of Optical subsystem
- X-X telecom subsystem offers lowcs1 cost among all three systems
- Optical communication performance improves over the other two systems as data rate is increased.
- High power SSPAS drive power requirements for X-X and Ka at 1 Gbit/day
- Acq/Trk subsystem sets threshold on power requirements of optical subsystem
    - Required onboard power for optical communication can be reduced by providing an uplink beacon.

The data on mass, Power-consumption and cost is compared in Figures IV-1 and IV-2 below.
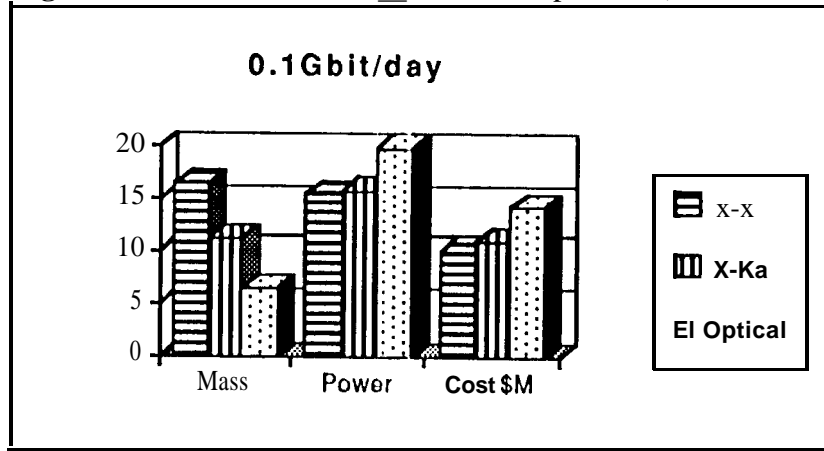
Figure IV-1 Mass, Power and Cost Comparison (O. 1 Gb/day

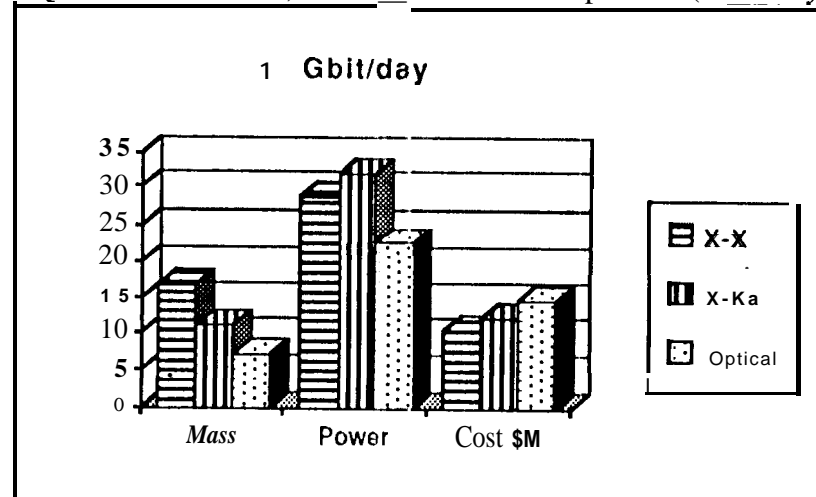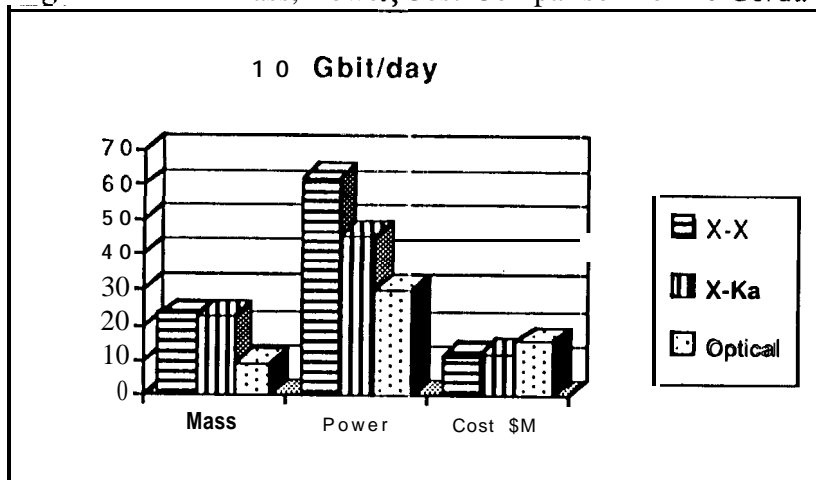**Figure** IV-1   Mass, Power and Cost Comparison (1 Gb/day

Figure IV-3.   Mass, Power, Cost Comparison for 10 Gb/day

2) 10 Gbits/day. For daily volume of 10 Gb/day, conclusions arc given below. Mass, power, and cost factors are compared in Figure IV-3.

. X-X power consumption is significantly higher than Ka or Optical
• Mass of Optical subsystem is 40% that of either X-Ka or X-X subsystems

V. FUTURE WORK

The **continuing** thrust of the study will be to refire our overall understanding of the relative merits of the X/X, X/Ka, and optical systems. To accomplish this goal, wc are seeking to minimize a cumulative figure of merit (FOM) for each technology over a series of Mars missions. This figurcaf-merit will include flight unit mass, power consumption, tracking costs, and NRE (non-recurring engineering), RE (recurring engineering) costs of developing flight and ground terminals. Minimizing this figure-of-merit will result in an optimal communication flight terminal and ground system configuration. The optimal payloads (X, Ka, and Optical) will then be compared with onc another.

Also, an alternative laser-communication flight system is being evaluated **assuming a** 3.5 m receiver telescope on the ground instead of a 10 m. Recommendations for future work arc to: Work on optical Acq/Trk issues to realize mass an power savings for daytime communications; Continue system studies support to identify areas of R&D in Ka-band and Optical technologies that impact future NASA deep-space missions; Develop an assessment of technology readiness for the flight and ground elements of the three pathways

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